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SUMMARY OF RESULTS OBTAINED BY TRANSONIC-BUMP METHOD ON EFFECTS OF PLAN FORM AND THICKNESS ON LIFT AND DRAG CHARACTERISTICS OF WINGS AT TRANSONIC SPEEDS

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SUMMARY

This paper presents a summary of the effects of plan form and thickness on the lift and drag characteristics of wings at transonic speeds. The data considered in this summary were obtained during a transonic research program conducted in the Langley high-speed 7- by 10-foot tunnel by the transonic-bump method in the Mach number range from 0.6 to 1.15. The Reynolds numbers of the tests were generally less than 1×10^6. The results indicated that, for subsonic Mach numbers below the force break, theoretical lift-curve-slope calculations were in fair agreement with the experimental results, whereas in the supersonic range the theoretical values were considerably higher than the experimental values. Increasing the thickness ratio caused rather large losses of lift in the transonic speed range and increasing the sweep angle decreased these losses. Decreasing the thickness ratio and increasing the sweep angle increased the drag-rise Mach number and reduced the pressure drag. The sonic pressure drag varied linearly with the 5/3 power of the thickness ratio in accordance with the two-dimensional transonic similarity rule. The effect of sweep angle on the maximum pressure drag could also be estimated with good accuracy. In general, the drag due to lift was increased by decreases in thickness ratio, increases in sweep angle, and decreases in aspect ratio.

INTRODUCTION

The National Advisory Committee for Aeronautics has been engaged in a transonic research program which was recommended by an NACA Special Subcommittee on Research Problems of Transonic Aircraft Design. As a part of this program a systematic investigation of the effects of wing thickness and plan form on the aerodynamic characteristics in the transonic range has been conducted in the Langley high-speed 7- by 10-foot tunnel by the transonic-bump method. In order to expedite the publication

[1]Supersedes declassified NACA RM L51H30, 1951.

of these data, the results for each wing were published separately and little analysis of the data was made.

The purpose of this paper is to compare the results obtained for the various wings of the transonic program in order to determine the effects of thickness and plan form on the lift and drag characteristics of wings in the transonic speed range. It was also desired to determine the extent to which the experimental results were predictable by available subsonic, transonic, and supersonic theories.

It should be pointed out that there are many shortcomings of the transonic-bump technique used to obtain the results presented in this paper. The Reynolds numbers are extremely low (see fig. 1), the spanwise Mach number gradients rather large, and the flow is slightly curved. However, the results are believed to give at least a qualitative indication of the type of compressibility effects that may be encountered in the transonic speed range and fairly reliable indications of trends in aerodynamic characteristics resulting from systematic changes in geometry.

SYMBOLS

C_L lift coefficient

C_D total drag coefficient

C_{D_o} drag coefficient at zero lift

$C_{D_{o(max)}}$ maximum drag coefficient at zero lift

C_D' sonic pressure drag $\left(C_{D_{o(M = 1.0)}} - C_{D_{o(M = 0.6)}}\right)$

$C_D'_{max}$ maximum pressure drag $\left(C_{D_{o(max)}} - C_{D_{o(M = 0.6)}}\right)$

ΔC_D drag due to lift $\left(C_D - C_{D_o}\right)$

α angle of attack, degrees

A aspect ratio

λ taper ratio

RESULTS AND DISCUSSION

The results are presented in the form of lift-curve slopes (measured through zero lift) and zero-lift drag coefficients against Mach number and drag due to lift against lift coefficient for several Mach numbers. Complete details of the basic data are presented in references 2 to 11 and are summarized in table I.

The present paper considers only wings having constant thickness ratios along the span. However, as an outgrowth of the original transonic research program, an investigation has been made to determine the effect of spanwise thickness variations on the aerodynamic characteristics of swept wings at transonic speeds and the results are presented in reference 12. Inasmuch as reference 12 contains a rather complete discussion of the results and includes comparisons with both the theoretical and the experimental results for wings of constant thickness ratios, the results are not reproduced here.

Lift

Effect of thickness ratio.- The effect of thickness ratio on the variation of the lift-curve slope with Mach number in the transonic speed range is shown in figure 2. In addition to the experimental lift-curve slopes, the theoretical critical Mach numbers as estimated from references 13 and 14 are presented as short vertical lines. Only the effects of airfoil section and wing sweep were considered in estimating the theoretical critical Mach numbers since reference 15 indicates that aspect ratio has a negligible effect in the aspect-ratio range (from 2 to 6) under consideration. Since the question of which sweep line is most directly related to the compressibility effects, especially for the lifting case, is somewhat controversial, the sweep angle of the wing reference line (quarter-chord line) was arbitrarily used to estimate the effect of sweep on the critical Mach number.

The top part of figure 2 shows the effect of thickness ratio on unswept wings of aspect ratio 4. The results indicate that, as the thickness ratio is increased, the force-break Mach number decreases as would be expected from theoretical critical Mach number considerations. However, it will be noted that the force-break Mach number occurs at a higher value than the theoretical critical Mach number. For the thicker wings there is a definite bucket-type variation of lift-curve slope with Mach number beyond the force break with the loss of lift in the bucket increasing with increasing thickness ratio. Although the 12-percent-thick wing was not of the same airfoil series, it is felt that the effects shown are due mainly to the maximum thickness ratio and not to the chordwise thickness distribution.

In the middle and lower parts of figure 2 the effect of thickness ratio on the variation of the lift-curve slope with Mach number for 45° sweptback wings of aspect ratios 4 and 6 is presented. The trends with thickness ratio for the swept wings are similar to those for the unswept wings, with the force break occurring somewhat beyond the theoretical critical Mach number and the loss in lift-curve slope in the transonic range increasing with increasing thickness ratio. In general, it can be said that, at moderate subsonic speeds, thickness ratio has relatively small effects on the lift-curve slope and that increasing thickness ratio reduces the force-break Mach number and increases the loss in lift-curve slope (shock stall) in the transonic speed range. At the higher Mach numbers, as the shock moves to the rear, lift is recovered on the thick wings so that, in general, there is relatively little effect of thickness ratio on the lift-curve slope.

Effect of aspect ratio.- The effect of aspect ratio on the variation of the lift-curve slope with Mach number in the transonic speed range is shown in figure 3. In addition to the experimental results, the theoretical results obtained from references 16 to 21 are presented. The upper part of the figure presents the effect of aspect ratio on the characteristics of an unswept wing having an NACA 65A004 airfoil section. The experimental as well as the theoretical results indicate that, as the aspect ratio is reduced, the variation of the lift-curve slope with Mach number is decreased and, although the force-break Mach number is not very well defined for the aspect-ratio-2 wing, it appears that reducing the aspect ratio from 4 to 2 increases the force-break Mach number by approximately 0.05. In the low subsonic range the experimental results are in fairly good agreement with the theoretical results; however, the experimental lift-curve slopes rise more rapidly with Mach number than do the theoretical slopes. In the supersonic range the theoretical lift-curve slopes are considerably higher than the experimental slopes. This difference is due, in part, to the fact that the theory is for infinitely thin wings and it has been shown in figure 2 that the thinner the wing the higher the supersonic lift-curve slope.

In the bottom part of figure 3 the effect of aspect ratio on a 35° sweptback wing having an NACA 65A006 airfoil parallel to the plane of symmetry is presented. Although reducing the aspect ratio from 6 to 4 reduced the magnitude of the lift-curve slope, it had very little effect on the variation of the lift-curve slope with Mach number. In the subsonic range the theoretical lift-curve slopes are somewhat lower than the experimental results, whereas in the supersonic range the theoretical results are considerably higher than the experimental results as was the case for the unswept wings.

Effect of sweepback.- The effect of sweepback on the variation of the lift-curve slope with Mach number in the transonic speed range is shown in figure 4. The top part of the figure presents the results for wings of aspect ratio 4, taper ratio of 0.6, and NACA 65A006 airfoil sections parallel to the plane of symmetry. Also presented are the theoretical results obtained from references 16 to 21 and the experimental results obtained in the Langley 6-inch supersonic tunnel (ref. 22).

The agreement between the experimental and theoretical lift-curve slopes in the subcritical Mach number range is fair, with the experimental slopes being somewhat higher. The higher experimental slopes may be due, in part, to the extremely low Reynolds numbers of the tests. It has been shown (see refs. 23 and 24) that, in the low Reynolds number range, the lift-curve slope increases with decreasing Reynolds number. The subsonic results indicate that, as the sweep angle is increased, the variation of the lift-curve slope with Mach number decreases and the force-break Mach number increases. In the transonic speed range, increasing the sweep angle eliminated the bucket-type variation of lift-curve slope, which occurred for the unswept wing. In the supersonic range the experimental results appear to fair reasonably well into the supersonic-tunnel results, and indicate, as does the supersonic theory, that the variation of lift-curve slope with Mach number decreases with increasing sweep angle. However, the magnitude of the experimental lift-curve slopes in the supersonic range is considerably less than that given by the theory, partially because of the fact that the wings were of finite thickness; whereas the theory is for infinitely thin wings. In addition to the data presented, there are data available for a 35° sweptback wing of $A = 4$ (see table I); but, to avoid confusion between the rather large number of curves, these data have been omitted from figure 4. The data for this wing, however, can be seen in figure 3.

In the lower part of figure 4 the effect of sweepback on the variation of the lift-curve slope with Mach number for wings of aspect ratio 6, taper ratio 0.6, and NACA 65A006 airfoil sections parallel to the plane of symmetry is shown. Except for the fact that the lift-curve slopes are somewhat higher because of the higher aspect ratio, the aspect-ratio-6 data indicate essentially the same effects of sweep as did the aspect-ratio-4 data.

Effect of fuselage.- Although the preceding discussion has been for wing-alone configurations, most of the wings have also been tested in combination with a fuselage with the wings positioned on the fuselage so that the quarter chord of the mean aerodynamic chord was at the point of maximum fuselage diameter. The addition of a fuselage increased the lift-curve slope in all cases except for the unswept wings where there was no appreciable effect. (See refs. 2 to 9.) However, there appears to be no consistent trend of the fuselage effect on the lift-curve slope. This inconsistency may be due to the fact that, as the wing plan form was changed, the position of the root chord with respect to the fuselage also changed or the fact that there was air leakage between the fuselage and the bump surface. Even if the air leakage did not alter the qualitative trends with wing geometry, it probably had a rather large effect on the magnitude of the fuselage effect. A few general observations can be made from the data, however. In general, the percent increase in the lift-curve slope, due to the fuselage, appeared to increase with wing sweep angle and usually was fairly constant with Mach number. For the cases where there appeared to be a variation with Mach number, it was usually an increase with increasing Mach number. The maximum increase obtained was about 25 percent.

Critical and force-break Mach numbers.- Figure 5 summarizes the effect of wing thickness ratio and sweep angle on the theoretical critical Mach number and the experimental force-break Mach number. The theoretical critical Mach numbers were estimated from references 13 and 14 (effect of aspect ratio can be neglected in range of aspect ratio under consideration), whereas the experimental force-break Mach numbers were taken as the Mach number at which C_{L_α} was a maximum. In the top part of figure 5 the effect of thickness ratio is summarized for the unswept and 45° swept wings. The results indicate that, as the thickness ratio is increased, the critical and force-break Mach numbers decrease. It will be noted, however, that the force-break Mach numbers exceed the theoretical critical Mach numbers by approximately 0.04 for the unswept wing and approximately 0.03 for the 45° sweptback wing.

The bottom part of figure 5 presents the effect of sweep angle on the theoretical critical Mach number and the experimental force-break Mach number for wings having NACA 65A006 airfoil sections parallel to the plane of symmetry. The solid line represents the theoretical variation of the critical Mach number with sweep angle whereas the symbols represent the experimental force-break Mach numbers. It will be noted again that the force-break Mach number exceeds the critical Mach number somewhat. However, when the variation of the critical Mach number with sweep angle was determined by using the force-break Mach number for the unswept wing as though it were the critical Mach number, fairly good agreement was obtained as indicated by the agreement between the dashed line (representing the modified theory) and the symbols. It therefore appears that, if the force-break Mach number of an unswept wing is known, the force-break Mach number for a swept wing having the same airfoil section parallel to the plane of symmetry can be estimated by the method of reference 14. It should be noted that the simple sweep theory as indicated by the long- and short-dash line greatly overestimates the effect of sweep on the critical Mach number. Based on the simple sweep theory, which states that only the flow normal to the sweep line affects the pressures, the critical Mach number is equal to the section critical Mach number times the reciprocal of the cosine of the sweep angle for the airfoil section normal to the sweep line. The reason for the failure of the simple sweep theory is that in the vicinity of the plane of symmetry of a swept wing, and at the wing tips, the flow cannot conform to the cosine rule.

Lift bucket.- In figures 2 and 4 it has been shown that some wings are characterized by a bucket-type variation of lift-curve slope with Mach number above the force break and it was noted that as the thickness ratio was increased the loss in lift-curve slope in the bucket increased. It was also noted that, as the sweep angle was increased, the loss in lift-curve slope was decreased. In figure 6 the loss in lift-curve slope in the bucket is plotted against thickness ratio (streamwise) for unswept and 45° sweptback wings of aspect ratio 4. The loss in lift-curve slope is defined as the difference between the maximum lift-curve slope and the value at the bottom of the bucket divided by the maximum. The results indicate that the loss in lift-curve slope increases approximately linearly with thickness ratio, with the unswept 12-percent-thick wing losing about 45 percent of its maximum lift-curve slope and the 45° sweptback 12-percent-thick wing losing about 40 percent. It should be remembered that the thickness ratios are in the stream direction and, therefore, the effectiveness of sweep in reducing the loss of lift is somewhat less than would be obtained if the sweep had been produced by rotation so that the thickness ratio normal to the sweep line was constant. It should also be pointed out that the Mach number gradient over the transonic bump could modify somewhat the magnitude of the lift loss; however, it is felt that the results give at least a qualitative indication of the effect of thickness and sweep on the loss of lift-curve slope.

Drag at Zero Lift

Effect of thickness ratio.- In figure 7 the effect of thickness ratio on the drag at zero lift is presented for unswept wings of aspect ratio 4 and 45° sweptback wings of aspect ratios 4 and 6. The results indicate that, as the thickness ratio is increased, the drag rise occurs at a lower Mach number, is steeper, and rises to a higher value. It will be noted that the wings are not all consistent with regard to airfoil section; however, it is believed that the differences in drag characteristics are due mainly to the maximum thickness ratio and not the chordwise thickness distribution.

In the top part of figure 8 the pressure drag for wings of aspect ratios 4 and 6 at a Mach number of 1.0 is plotted against the transonic-similarity-rule thickness parameter. (See ref. 25.) The pressure drag at a Mach number of 1.0 was assumed to be the difference between the total drag at a Mach number of 1.0 and the total drag at a Mach number of 0.6. The results indicate that, for wings of moderate aspect ratio, the sonic pressure drag varies fairly linearly with $(t/c)^{5/3}$ even for the swept wings, although the theory is for two-dimensional unswept wings. It will be noted, however, that when the sweep angle was increased to 45° the variation of pressure drag with $(t/c)^{5/3}$ decreased. In the bottom part of figure 8 the experimental effect of aspect ratio on the variation of sonic pressure drag with $(t/c)^{5/3}$, as determined from references 26, 27, and 28, is presented for unswept wings having NACA 65-0XX airfoil sections. Also presented is the result of the present tests and it will be noted that the pressure drag of the present tests is somewhat higher than that obtained from references 26, 27, and 28. The difference in drag may be due to the extremely low Reynolds number of the present tests or wing-fuselage interference which might be present in the data of references 26, 27, and 28. Although it is for a different type of airfoil section, the theoretical result of reference 29 is presented since it is one of the few sonic solutions available.

Effect of sweepback.- The effect of sweepback on the drag at zero lift is presented in figure 9. As the angle of sweepback is increased, the drag rise becomes less steep and begins at a higher Mach number because of the effectiveness of sweep in reducing the pressure drag. No transonic theories appear to be available to predict the effect of sweep on the drag at zero lift. However, the maximum pressure-drag coefficient, which usually occurs in the transonic range, has been found to decrease by the 4th power of the cosine of the sweep angle (ref. 30) when the thickness ratio is constant in planes normal to the sweep line (sweep obtained by rotation). The wings of the present paper, however, were swept by shearing the sections back, and the thickness ratio is therefore constant in planes parallel to the plane of symmetry. Therefore, the maximum pressure-drag coefficient of an unswept wing having a thickness ratio equal to the thickness ratio of the swept wing in a direction normal to the sweep line must be known before the maximum pressure-drag coefficient of the swept wing can be estimated. Figure 8 shows that the pressure drag for unswept wings at sonic speed is proportional to the 5/3 power of the thickness ratio and figure 7 indicates that the maximum pressure drag for thick unswept wings occurs very close to sonic velocity. Inasmuch as most swept wings are rather thick in the direction normal to the sweep line, it appears that the relationship for the pressure drag of unswept wings at the speed of sound can be used to estimate the maximum pressure-drag coefficient of sweptback wings. The expression for the maximum pressure drag then becomes

$$C_{D}'_{max} = K\left(\frac{t/c}{\cos \Lambda}\right)^{5/3} \cos^{4}\Lambda \qquad (1)$$

where

$$K = \frac{\partial C_D{}'}{\partial\left(\frac{t}{c}\right)^{5/3}}$$

(for unswept wing having same aspect ratio as swept wing) and $\frac{t}{c}$ is measured parallel to the plane of symmetry. Figure 10 shows a comparison of equation (1) with the experimental results obtained from figures 7 and 9. The top part of figure 10, using a value of 2.89 for K as obtained from figure 8, presents the results for the aspect-ratio-4 wings. The bottom part presents the results for the aspect-ratio-6 wings with a value of 3.27 for K, obtained by correcting the value for the aspect-ratio-4 wings of the present tests with aid of the aspect-ratio curve presented in figure 8. It will be noted that the agreement between equation (1) and the experimental results is good for both the aspect-ratio-4 and aspect-ratio-6 wings. Inasmuch as there is very little difference in the sweep of the reference line (quarter-chord line) and the sweep of the maximum-thickness line, and since in the transonic range the minimum pressure may not occur at the maximum thickness, the sweep of the reference line was used. The good agreement, therefore, may be due to the fact that the wings were not highly tapered.

Drag Due to Lift

Effect of thickness ratio.- The effect of wing thickness ratio on the drag due to lift is shown in figures 11 and 12. In figure 11 the drag-due-to-lift increment ΔC_D is plotted against lift coefficient for unswept wings having aspect ratio 4 and thickness ratios of 4, 6, and 12 percent at Mach numbers of 0.70, 0.95, and 1.15. The results indicate that, at a Mach number of 0.70, the drag due to lift increased as the thickness ratio was decreased, whereas at Mach numbers of 0.95 and 1.15 the reverse was true. In the lower right-hand part of the figure the drag due to lift for a lift coefficient of 0.30 is plotted against thickness ratio for Mach numbers of 0.70 and 1.15. Also presented are the theoretical values for the condition of the resultant force acting normal to the local relative wind $C_L^2/\pi A$ and the condition of the resultant force acting normal to the chord line $C_L \tan \alpha$. It will be noted that, at a Mach number of 0.70, there is a transition of the experimental values from $C_L \tan \alpha$ to $C_L^2/\pi A$ as the thickness ratio is increased from 4 percent to 12 percent. This transition may be due to the fact that the thin wing, which has a relatively sharp leading edge, may lose (because of leading-edge separation) a large part of its leading-edge suction. With zero leading-edge suction the resultant force is normal to the chord line. At a Mach number of 1.15 it will be noted that the drag due to lift is equal to $C_L \tan \alpha$. This condition is to be expected since these wings are unswept and therefore have supersonic leading edges and can develop no leading-edge suction.

In figure 12 the drag due to lift is plotted against lift coefficient for 45° sweptback wings having aspect ratio 6 and thickness ratios of 6, 9, and 12 percent at Mach numbers of 0.70, 0.95, and 1.15. As was the case for the unswept wings (fig. 11), the drag due to lift at a Mach number of 0.70 increases with decreasing thickness ratio. At a Mach number of 0.95 the drag due to lift was greatest for the thickest wing, whereas at a Mach number of 1.15 there was little effect of thickness ratio on the drag due to lift. In the lower right-hand part of figure 12 the drag due to lift at a lift coefficient of 0.40 is plotted against thickness ratio for Mach numbers of 0.70 and 1.15. As was the case for the unswept wings, there is a transition from the resultant force acting normal to the chord $C_L \tan \alpha$ to the resultant force acting normal to the local relative wind ${C_L}^2/\pi A$ as the thickness is increased from 6 percent to 12 percent. As mentioned previously, this transition may be due to leading-edge separation on the thin wings. At a Mach number of 1.15 it will be noted, as would be expected, that these swept wings with subsonic leading edges maintain some suction and do not have their resultant forces normal to the chord as did the unswept wings (fig. 11).

Effect of aspect ratio.- The effect of wing aspect ratio on the drag due to lift is shown in figure 13. The wings had aspect ratios of 4 and 2, were unswept, and had NACA 65A004 airfoil sections. The results indicate, as would be expected, that throughout the Mach number range investigated the aspect-ratio-2 wing has considerably more drag due to lift than the aspect-ratio-4 wing. In the lower right side of figure 13 the drag due to lift at a lift coefficient of 0.40 is plotted against aspect ratio and compared with ${C_L}^2/\pi A$ and $C_L \tan \alpha$ for Mach numbers of 0.70 and 1.15. The results for a Mach number of 0.70 indicate that the inclination of the resultant force approaches the normal to the chord line $C_L \tan \alpha$. This inclination may be due to a loss in leading-edge suction due to leading-edge separation on these thin wings (4 percent) with small leading-edge radii. The results for a Mach number of 1.15 indicate, as would be expected since the wing has a supersonic leading edge at this Mach number, that the resultant force is normal to the wing chord line.

Effect of sweepback.- The effect of wing sweep angle on the drag due to lift is shown in figure 14. In this figure the drag-due-to-lift increment ΔC_D is plotted against lift coefficient for wings of aspect ratio 4 and taper ratio 0.6 with NACA 65A006 airfoil sections parallel to the plane of symmetry and with sweep angles of 0°, 45°, and 60° at Mach numbers of 0.70, 0.95, and 1.15. The results indicate that, at Mach numbers of 0.70 and 0.95, the drag due to lift increases as the sweep angle increases, the drag due to lift of the 60° wing being about twice that of the unswept wing. However, at a Mach number of 1.15 there is very little effect of sweep angle on the drag due to lift. The reason for the increase in drag due to lift with sweep in the subsonic range is illustrated in the lower right-hand part of figure 14 where the experimental ΔC_D and the two theoretical curves - one for the resultant force acting normal to the local relative wind $C_L^2/\pi A$ and the other for the resultant force acting normal to the chord $C_L \tan \alpha$ - are presented for $C_L = 0.35$. It will be noted ($M = 0.70$) that, for the unswept wing, the resultant force appears to be about halfway between the normal to the local relative wind and the normal to the chord line; therefore, the induced drag depends not only on the induced angle but also on the geometric angle of attack. Since a swept wing requires a higher angle of attack to support a given lift (see fig. 4), it follows that the induced drag will increase with sweep angle. The reason for the rearward inclination of the resultant force is probably caused, to a large extent, by leading-edge separation on these thin (6 percent) wings with small leading-edge radii. This leading-edge separation results in a loss of leading-edge suction which corresponds to an increase in drag. With zero leading-edge suction the resultant force is normal to the chord line (neglecting any separation rearward of the leading edge) and the induced drag is given by $C_L \tan \alpha$. It will also be noted on the plot of ΔC_D against sweep angle ($M = 0.70$) that the 60° wing apparently has lost more suction than the unswept wing. This loss may be due to the fact that, according to simple sweep theory, the lift coefficient based on the component of the dynamic pressure normal to the sweep line is important and this lift coefficient increases with sweep angle for a constant wing lift coefficient, based on the free-stream dynamic pressure.

A possible explanation of the fact that the drag due to lift is relatively independent of sweep angle at a Mach number of 1.15 can be seen from the plot of ΔC_D against sweep angle ($M = 1.15$) in the lower right-hand corner of figure 14. It appears that the unswept wing has lost a large part of its leading-edge suction due to the fact that at this Mach number it has a supersonic leading edge. However, the swept wings have subsonic leading edges and, therefore, retain a part of their leading-edge suction which results, for these particular wings at this particular Mach number, in a rather flat curve of ΔC_D against sweep angle.

Effect of fuselage.- The effect of a fuselage on the drag due to lift of a 45° swept wing of aspect ratio 4 with an NACA 65A006 airfoil section parallel to the plane of symmetry is presented in figure 15. In this figure the drag due to lift ΔC_D is plotted against lift coefficient for Mach numbers of 0.70, 0.95, and 1.15 for both the wing alone and the wing-fuselage combination. It is interesting to note that the drag due to lift of the wing-fuselage combination is considerably less than that for the wing alone. This difference is due, at least in part, to the fact that on this relatively thin wing the drag due to lift is dependent to a rather large extent on the angle of attack, and the addition of the fuselage increases the lift-curve slope and thereby reduces the drag at a given lift coefficient.

CONCLUSIONS

A correlation, based on transonic-bump data, of the effect of wing thickness ratio and plan form on the lift and drag characteristics in the Mach number range from 0.6 to 1.15 at Reynolds numbers generally lower than 1×10^6 indicated the following conclusions:

1. In the subsonic range the theoretical results were in fair agreement with the experimental lift-curve slopes below the force break, but in the supersonic range the theoretical results were considerably higher than the experimental results.

2. Increasing the thickness ratio caused rather large losses in lift-curve slope in the transonic speed range and increasing the sweep angle decreased these losses. The effect of sweep angle on the lift-force-break Mach number could be estimated with a fair degree of accuracy by utilizing critical Mach number theory.

3. Increasing the thickness ratio caused an earlier drag rise and large increases in the pressure drag. The results indicated that, at a Mach number of 1.0, the pressure drag was approximately porportional to the 5/3 power of the thickness ratio.

4. Increasing the sweep angle increased the drag-rise Mach number and reduced the pressure drag. The effect of sweep on the maximum zero-lift drag could be estimated fairly accurately by a previously determined relationship.

5. In the subcritical Mach number range, decreases in thickness ratio caused increases in the drag due to lift, probably because of leading-edge separation. However, in the supercritical range the opposite was

generally true, with the undesirable effects of thickness on the lift-curve slope being reflected in the drag due to lift.

6. Decreasing the aspect ratio caused increases in the drag due to lift throughout the Mach number range.

7. Increasing the sweep angle caused an increase in the drag due to lift in the subsonic range but had little effect at a Mach number of 1.15.

8. The addition of a fuselage caused a reduction in the drag due to lift for cases where the lift-curve slope was increased by the fuselage.

Langley Aeronautical Laboratory,
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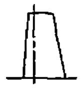
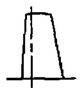
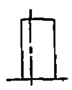
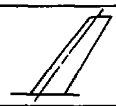
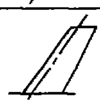
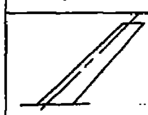
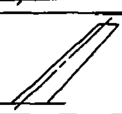
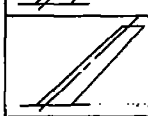
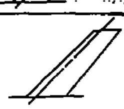
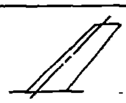
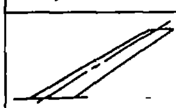
TABLE I

Summary of Wing Geometry

	Λ	A	λ	Section	Ref.	Remarks
	0	4.0	0.60	65A004	2	
	0	4.0	0.60	65A006	3	
	0	4.0	1.00	0012	—	
	0	2.0	0.78	65A004	10	
	35	6.0	0.60	65A006	4	
	35	4.0	0.60	65A006	5	C_{D_0} presented in reference 4
	45	6.0	0.60	65A006	6	
	45	6.0	0.60	65A009	7	
	45	6.0	0.56	63_1A012	11	
	45	4.0	0.60	65A006	8	C_{D_0} presented in reference 6
	45	4.0	0.68	63_1A012	11	
	60	4.0	0.60	65A006	9	

NACA

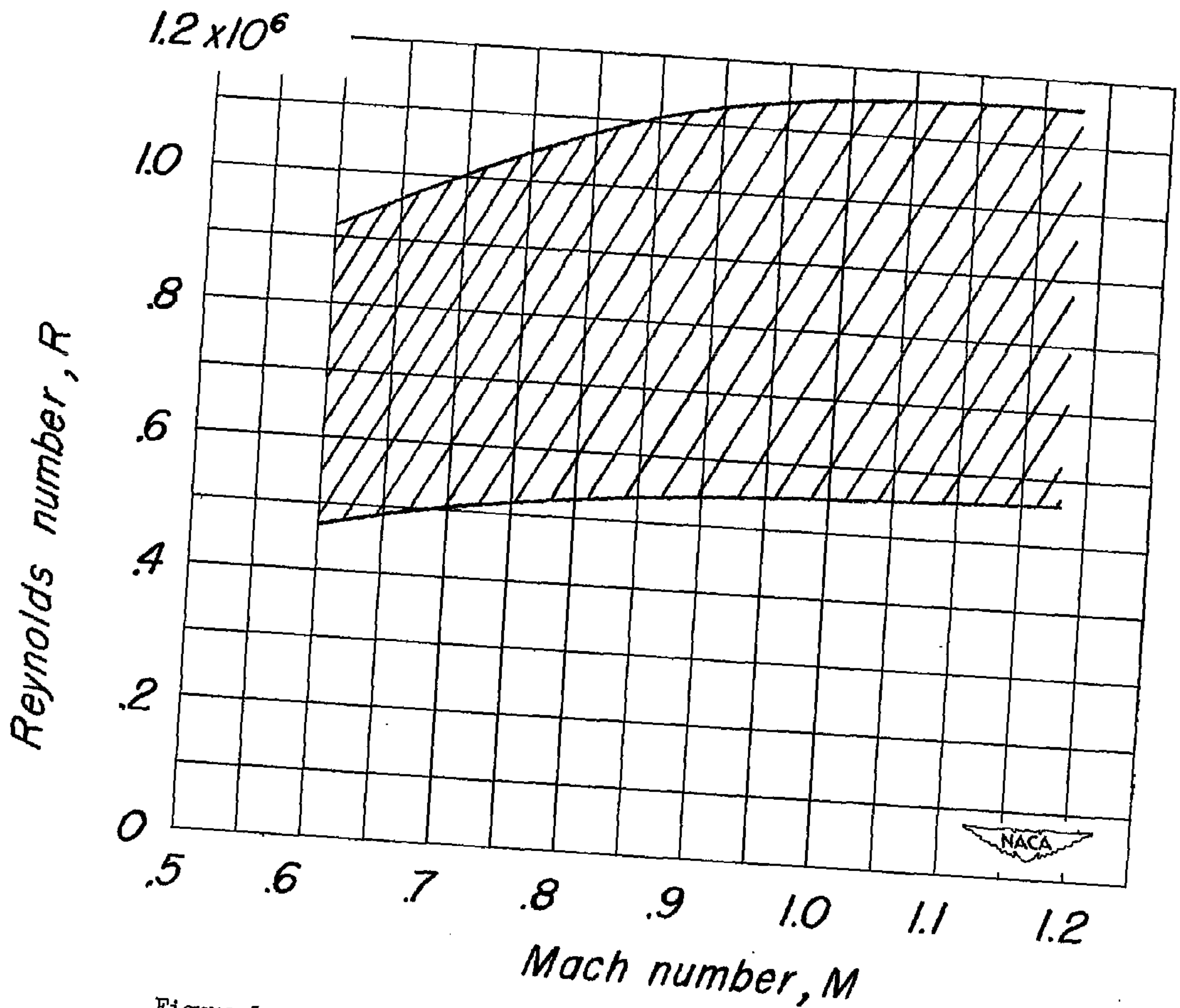


Figure 1.- Reynolds number range of wings used in correlation.

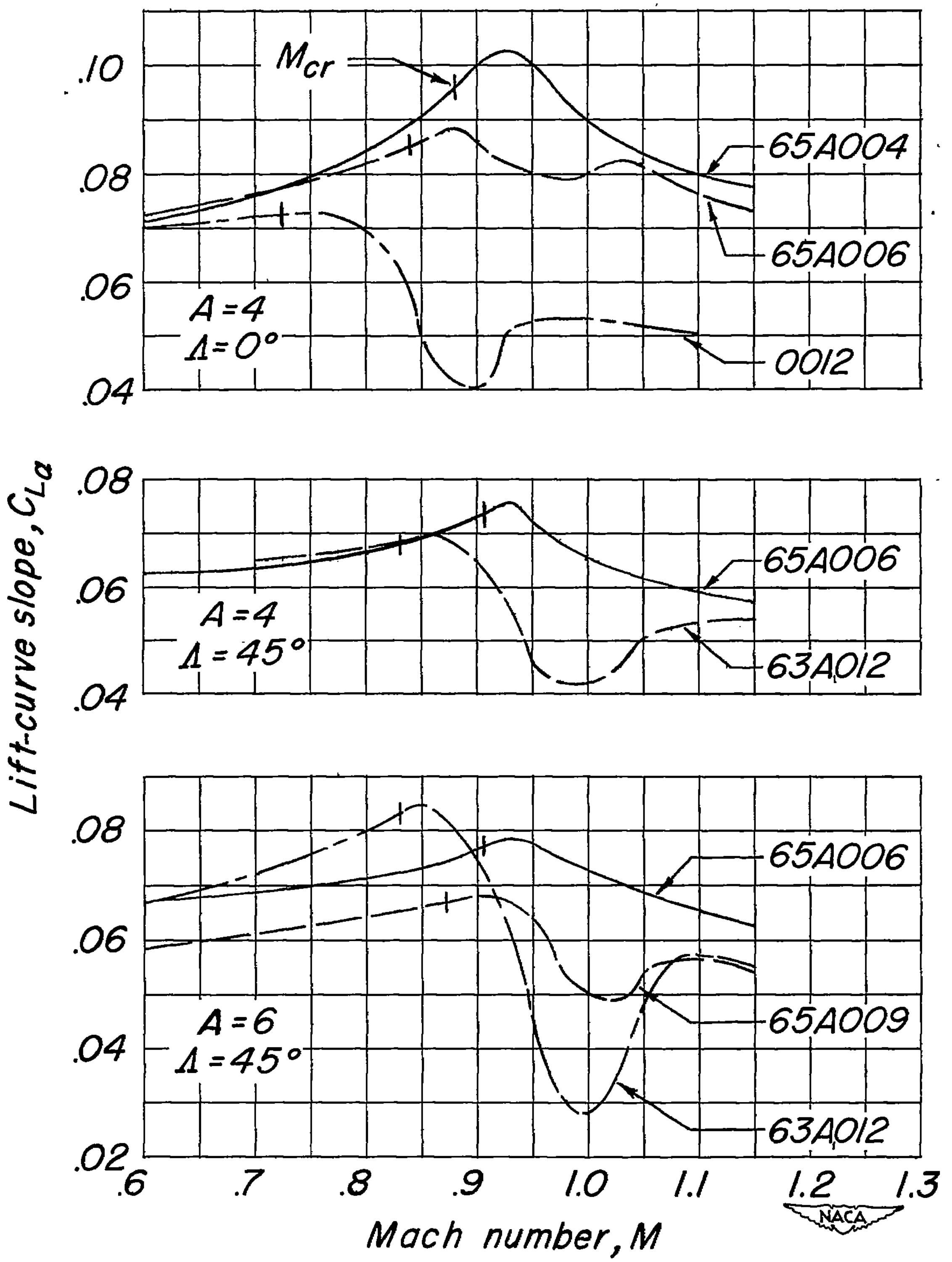


Figure 2.- Effect of thickness ratio on the variation of lift-curve slope with Mach number.

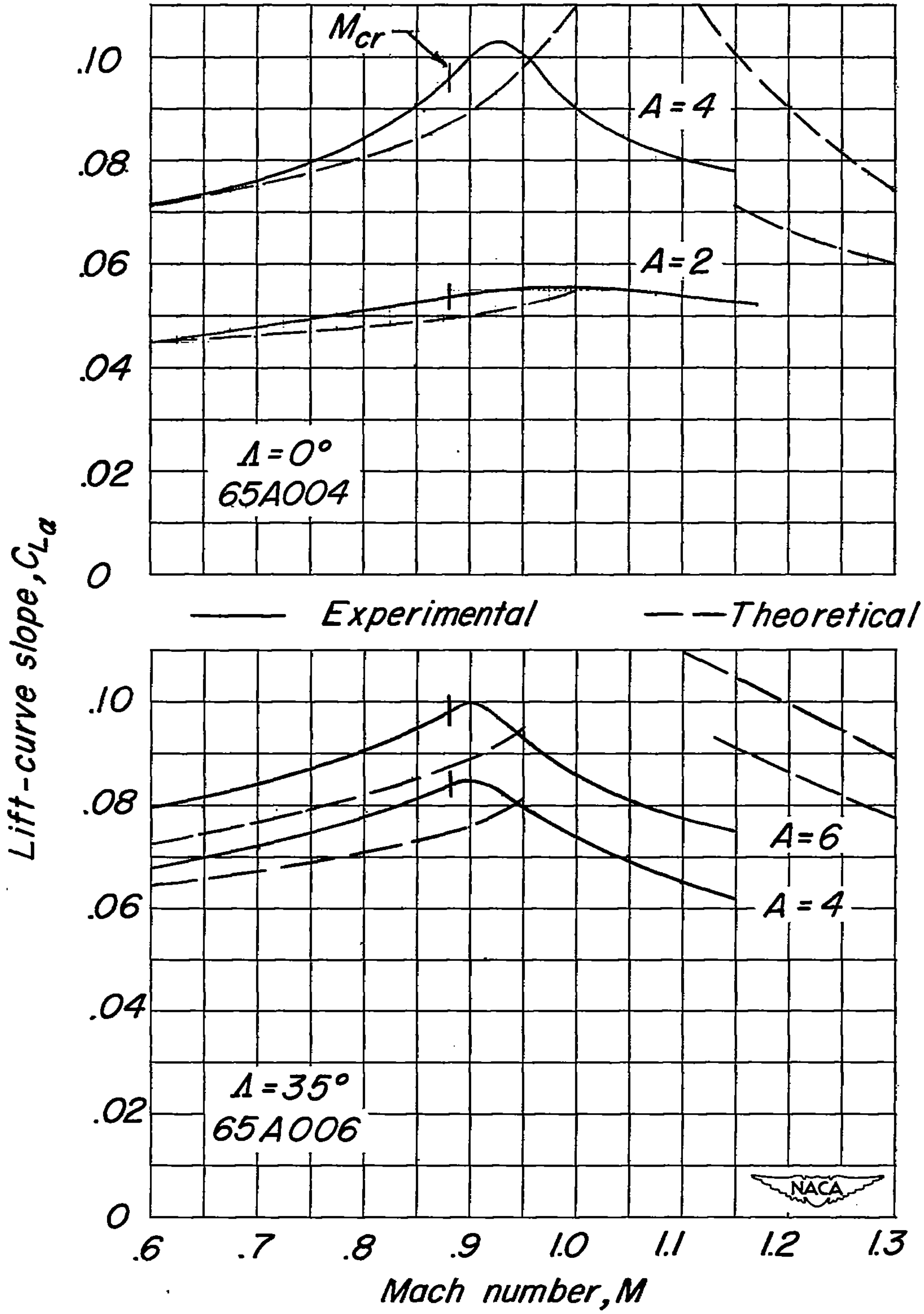


Figure 3.- Effect of aspect ratio on the variation of lift-curve slope with Mach number.

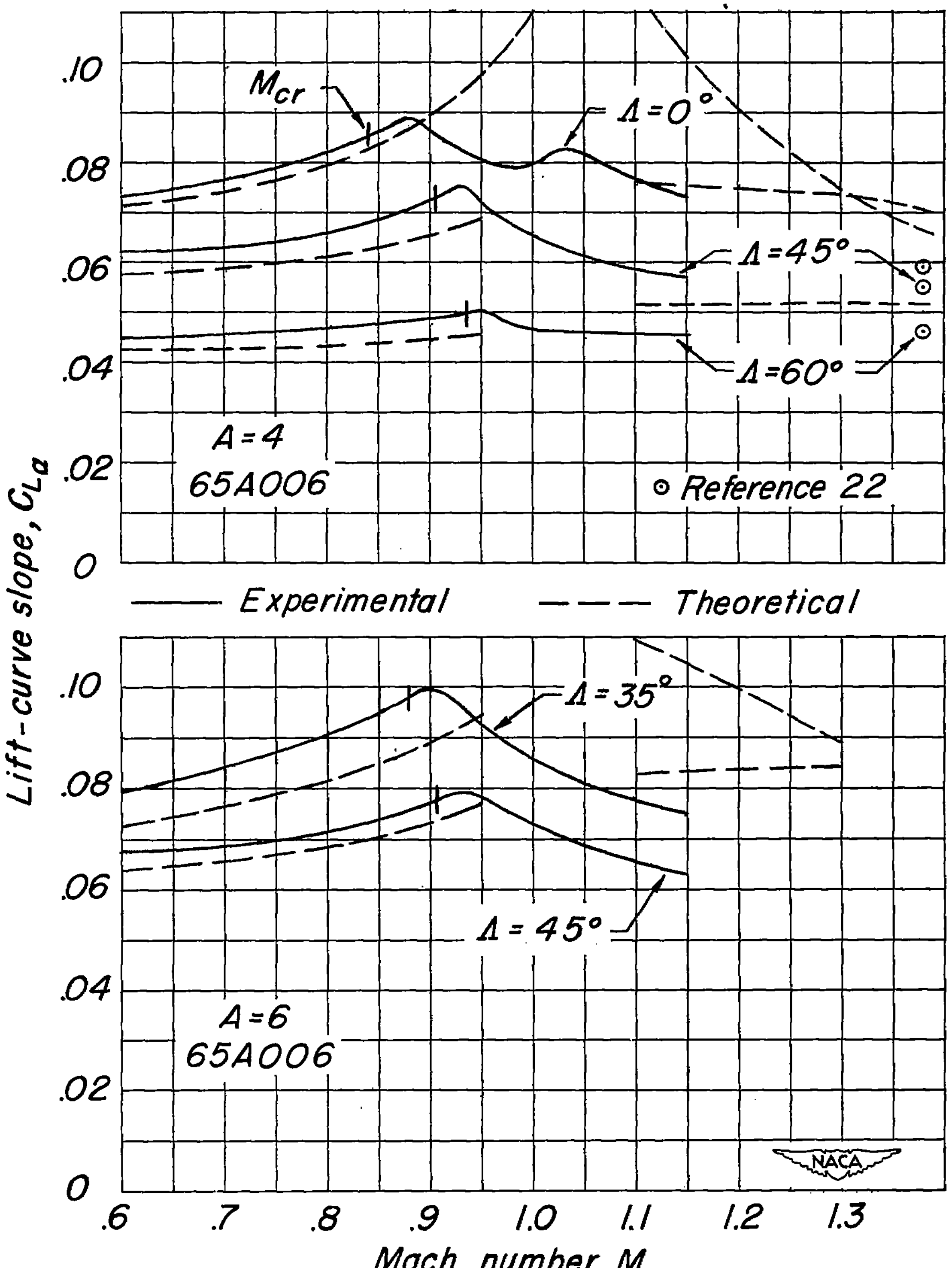
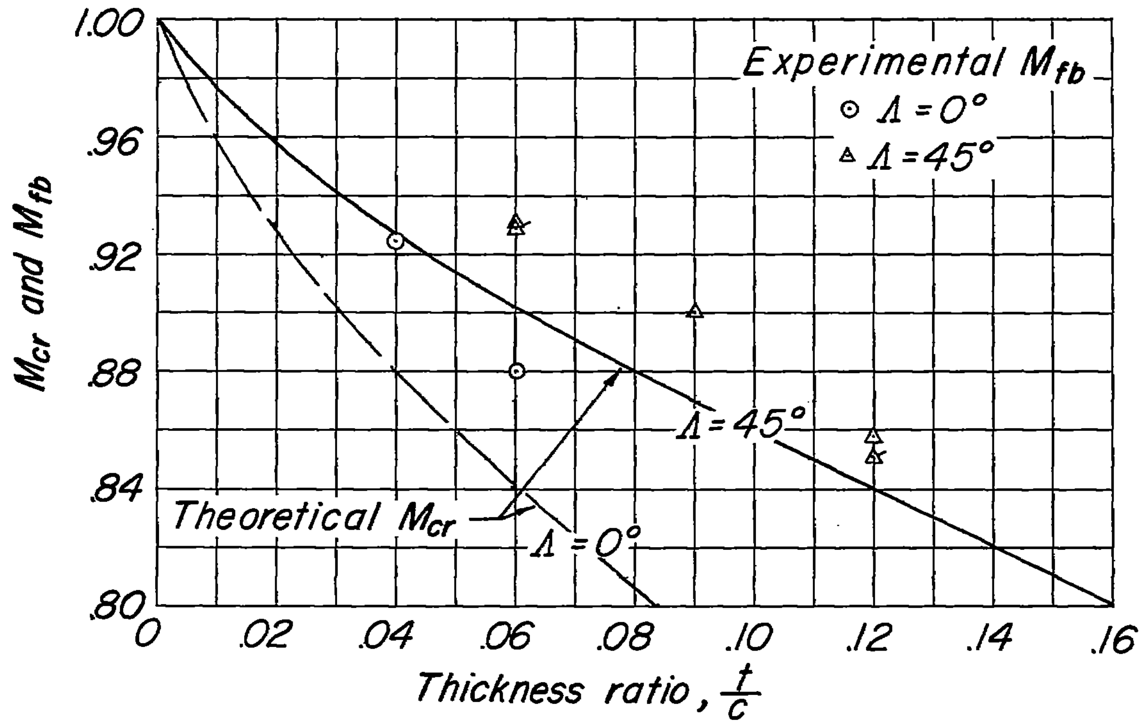


Figure 4.- Effect of sweep angle on the variation of lift-curve slope with Mach number. $\lambda = 0.6$.



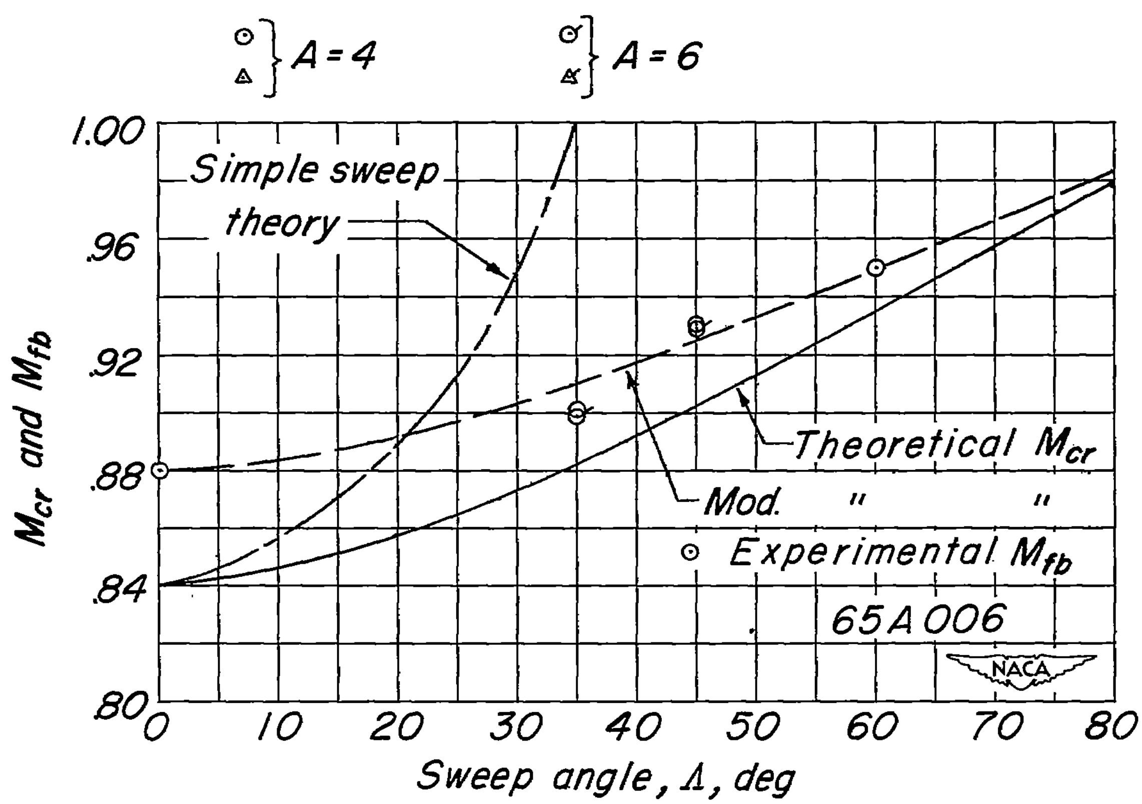


Figure 5.- Effect of thickness ratio and sweep angle on M_{cr} and M_{fb}.

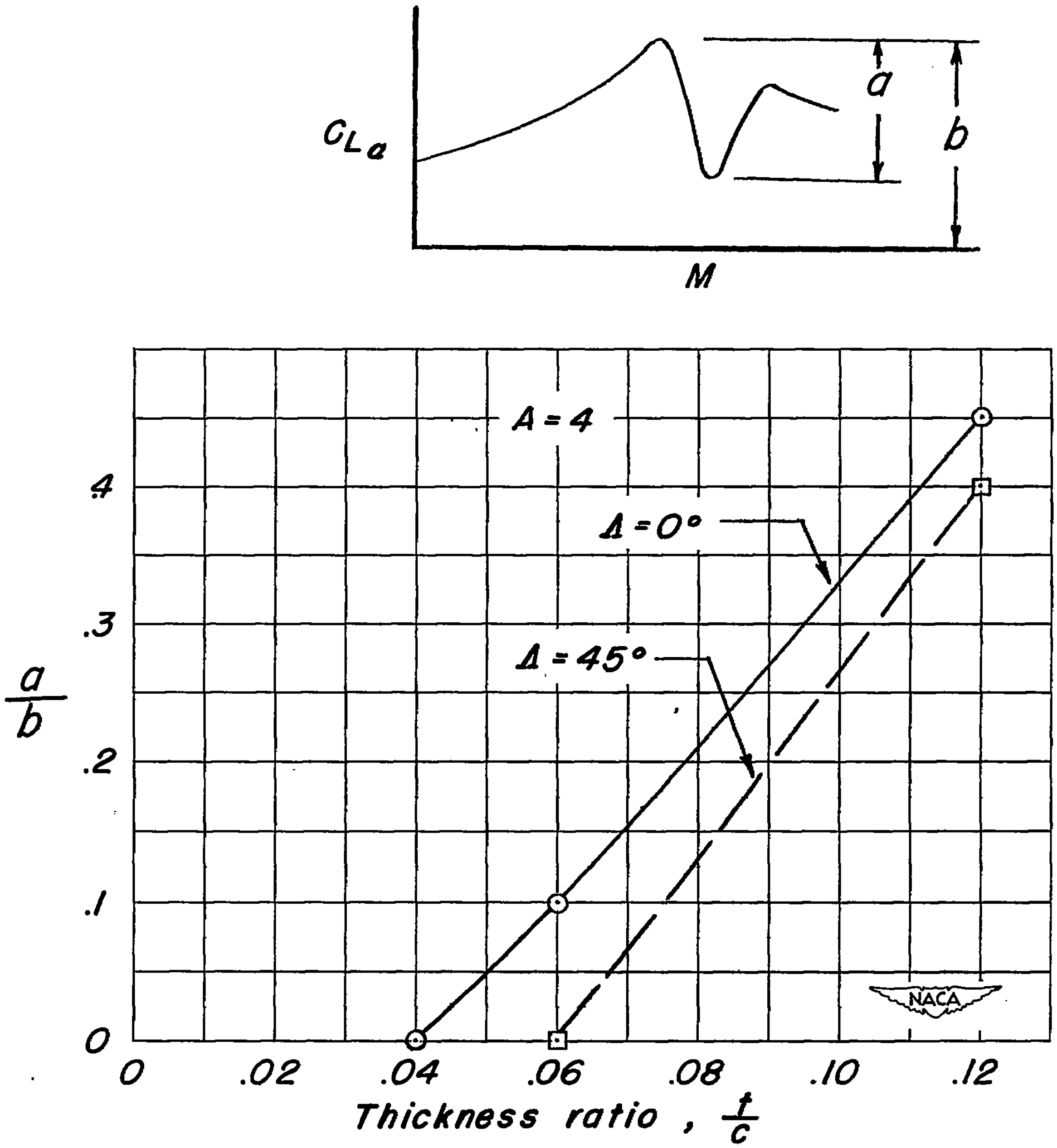


Figure 6.- Effect of thickness ratio and sweep angle on the transonic lift bucket.

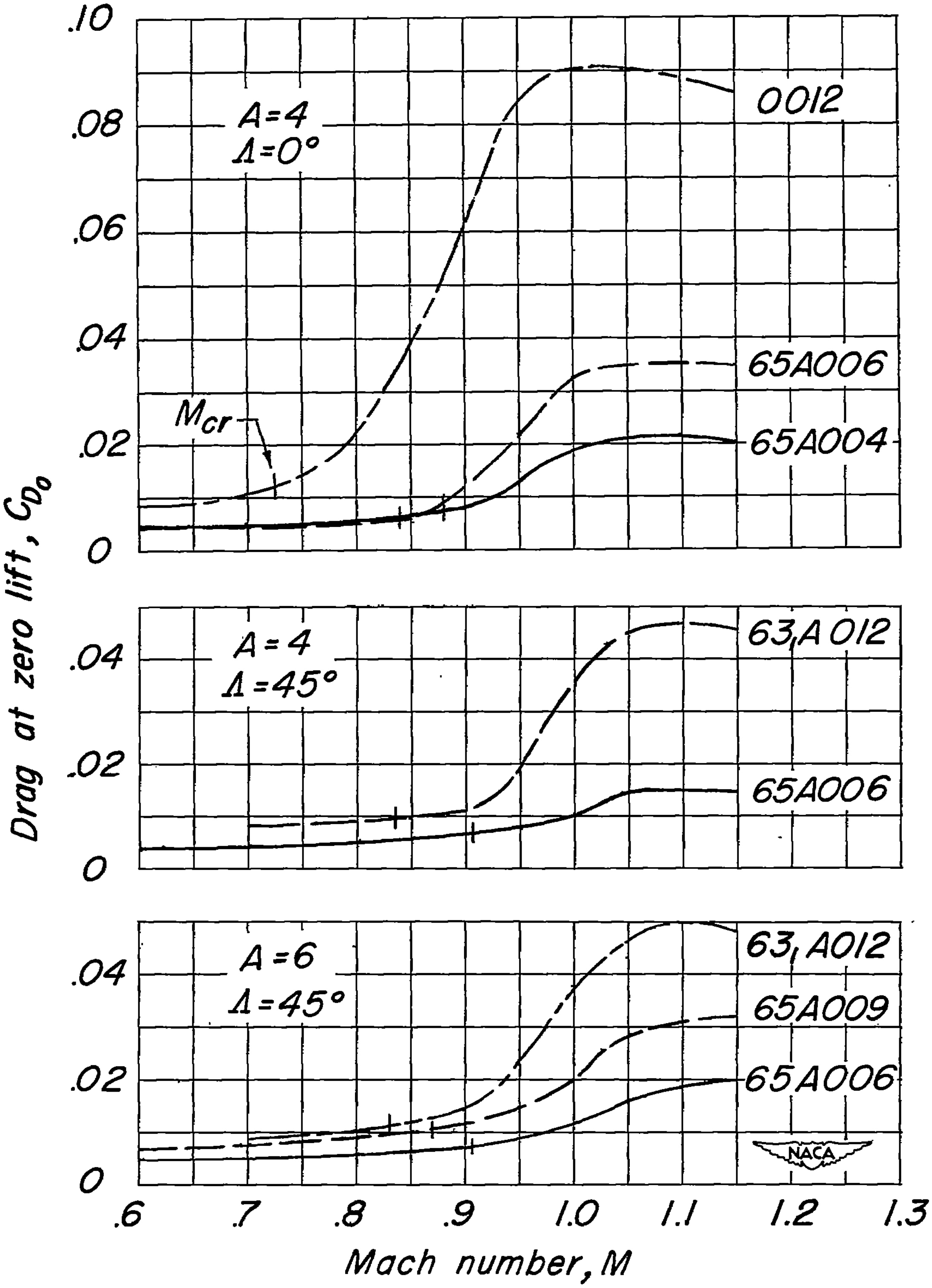


Figure 7.- Effect of thickness ratio on the drag at zero lift.

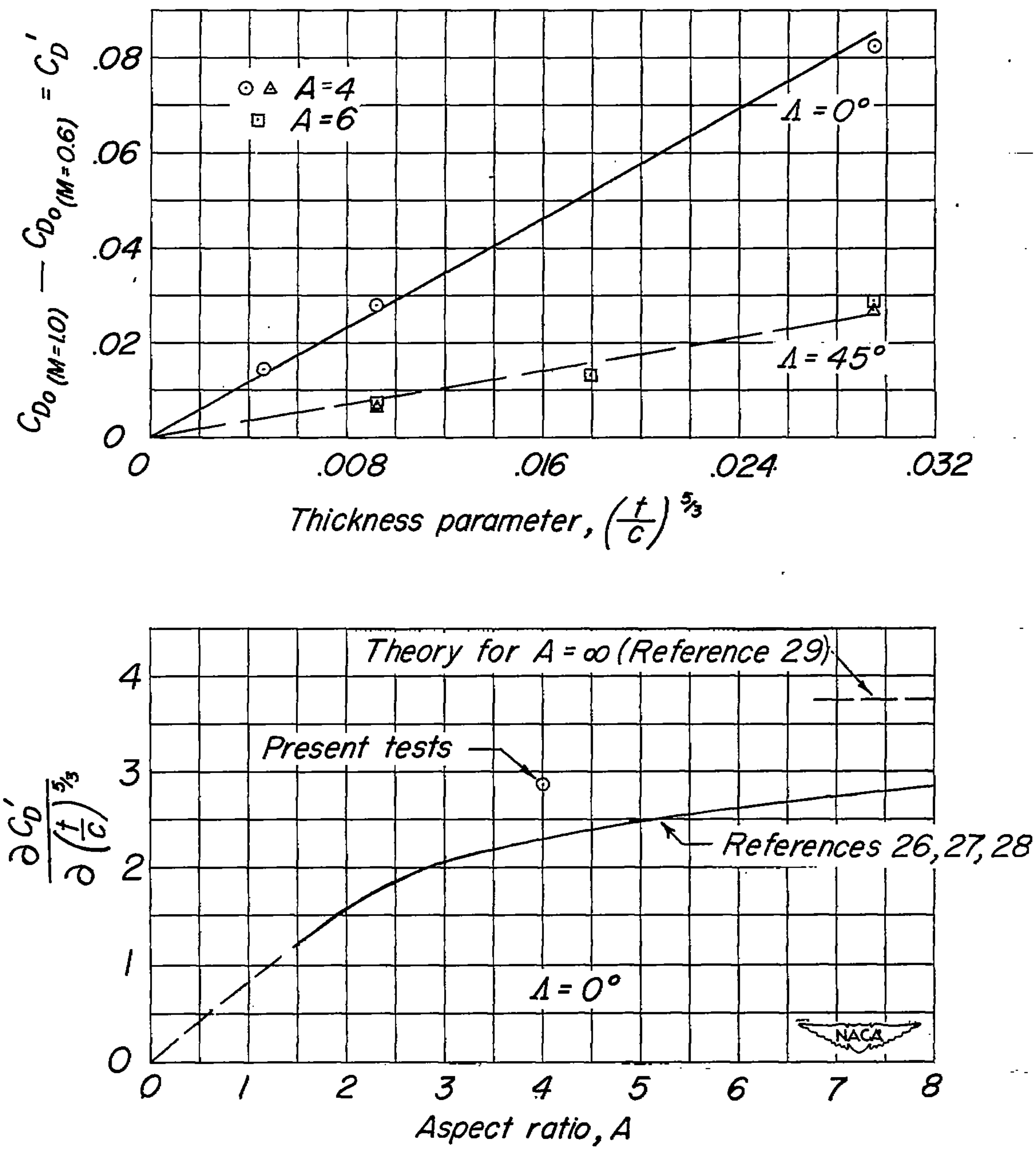


Figure 8.- Effect of thickness ratio on the sonic pressure drag.

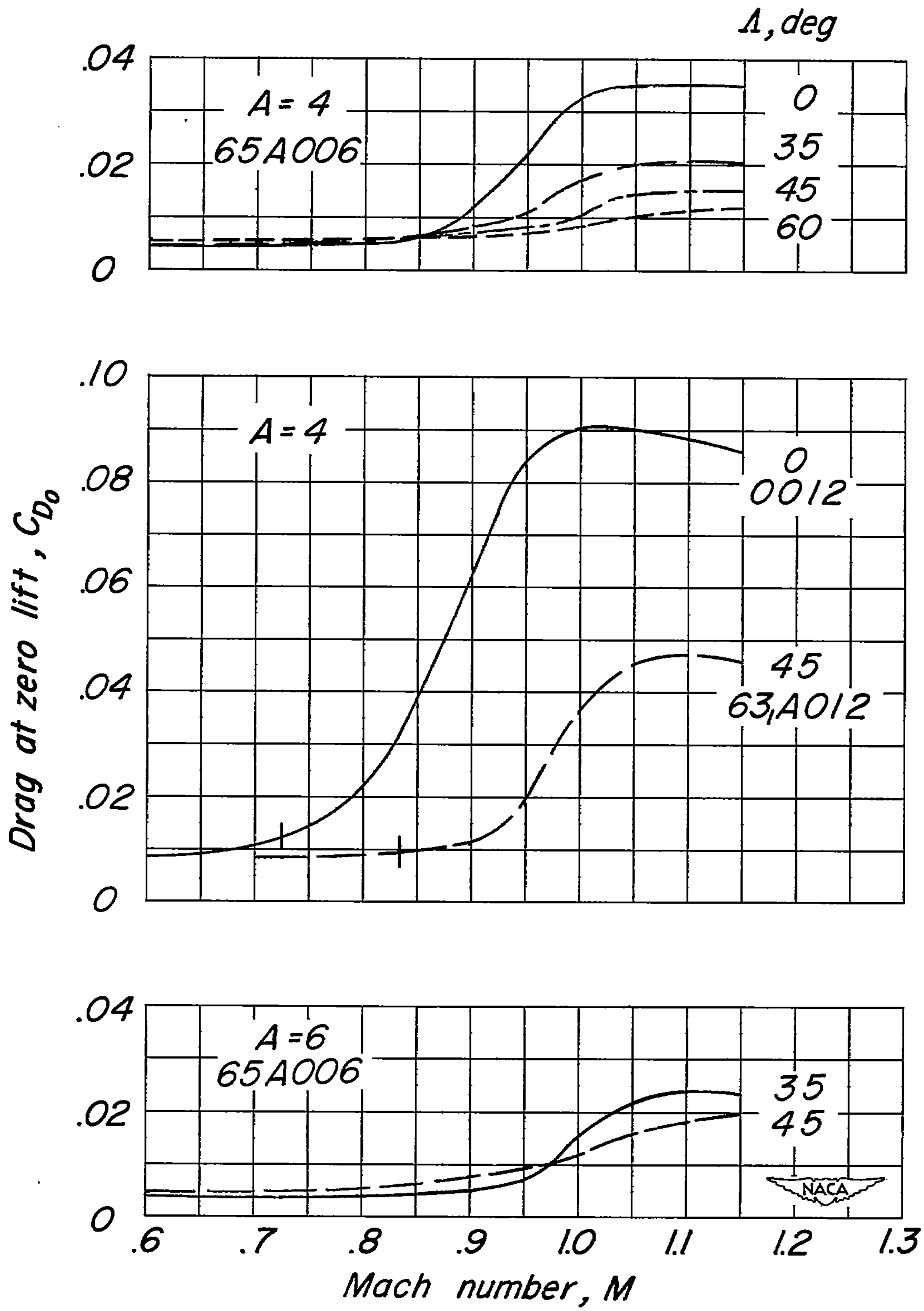


Figure 9.- Effect of sweep angle on the drag at zero lift.

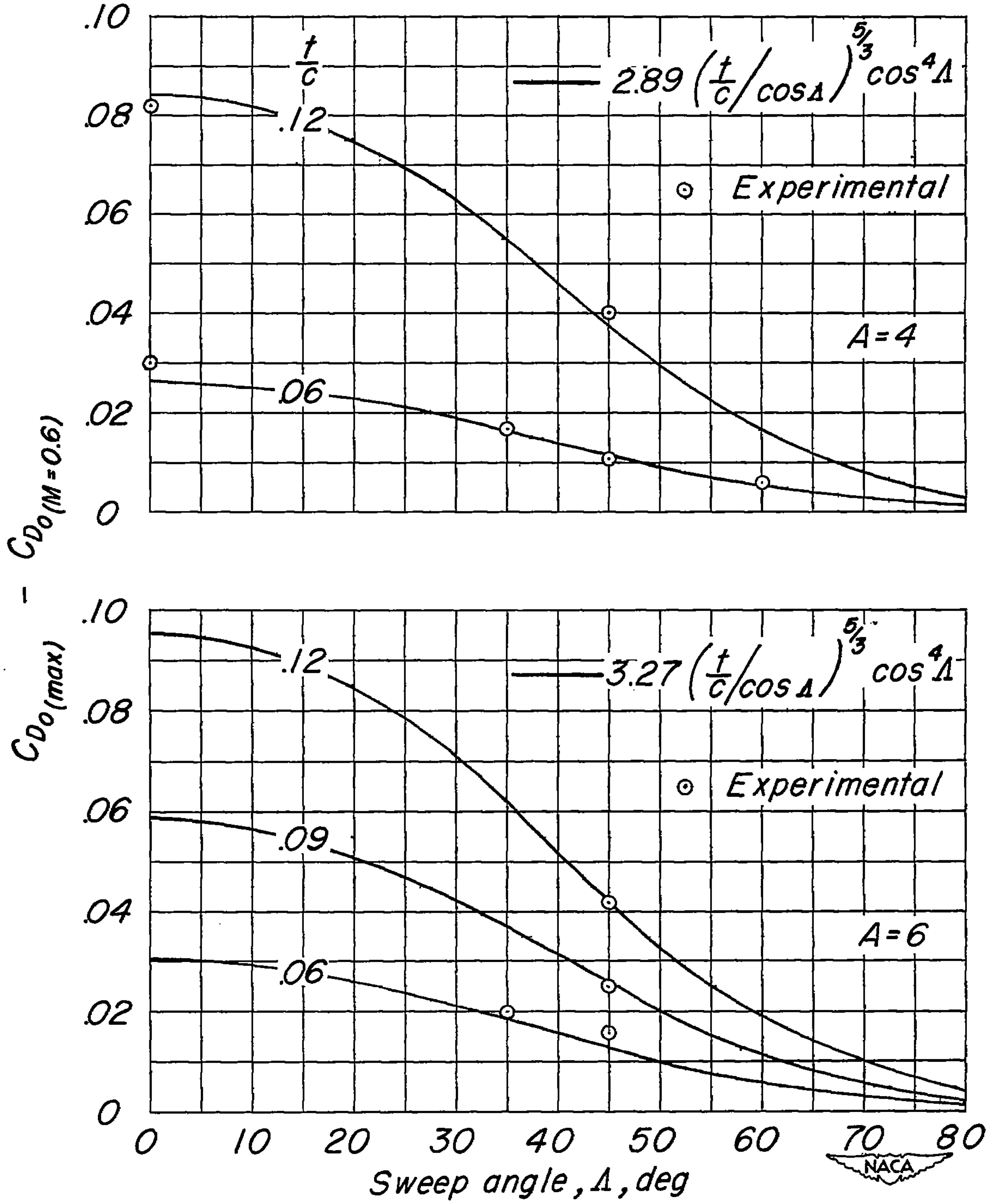


Figure 10.- Effect of sweep angle on the maximum transonic drag at zero lift.

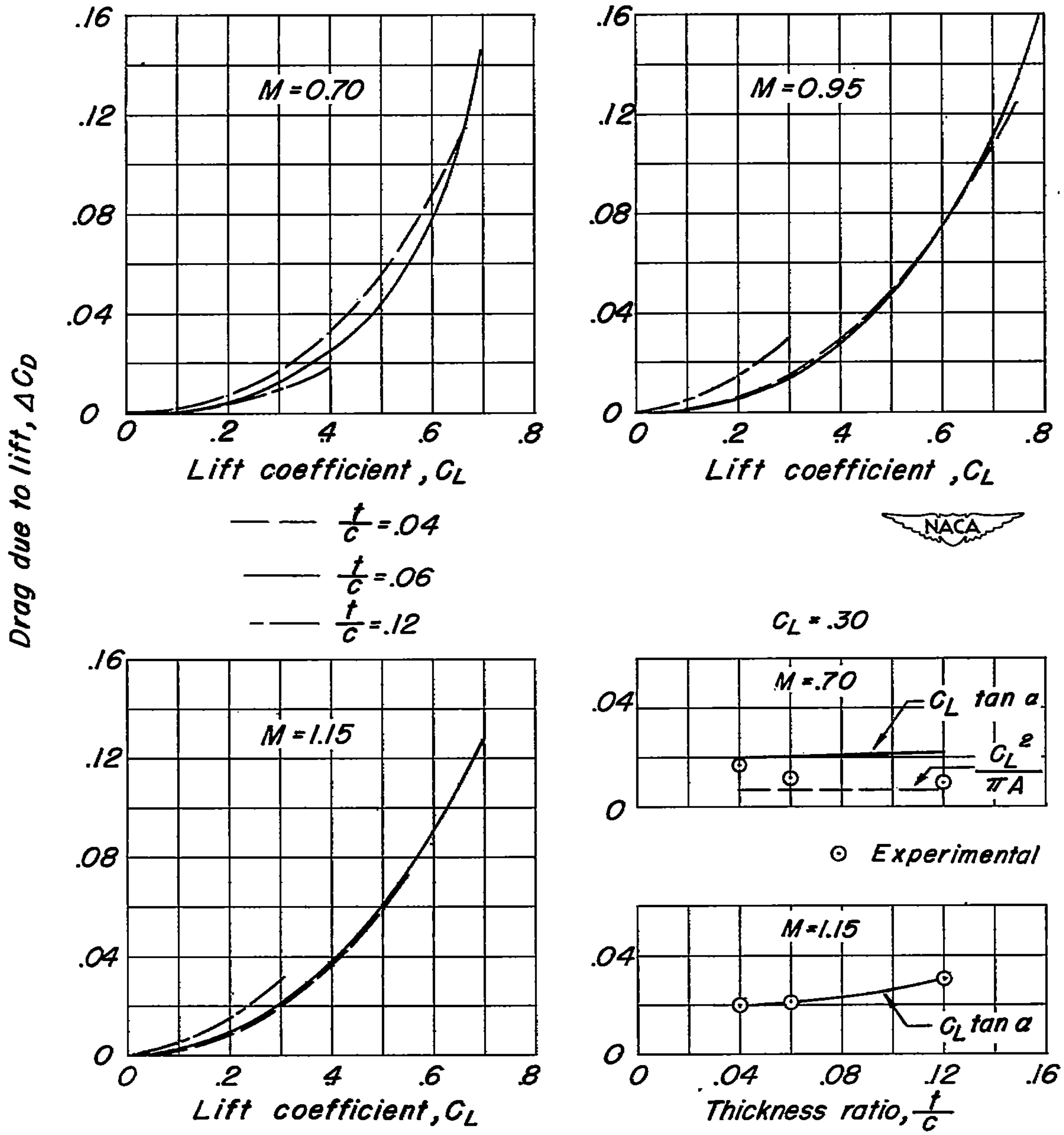


Figure 11.- Effect of thickness ratio on the drag due to lift. $A = 4$; $\Lambda = 0^{\circ}$.

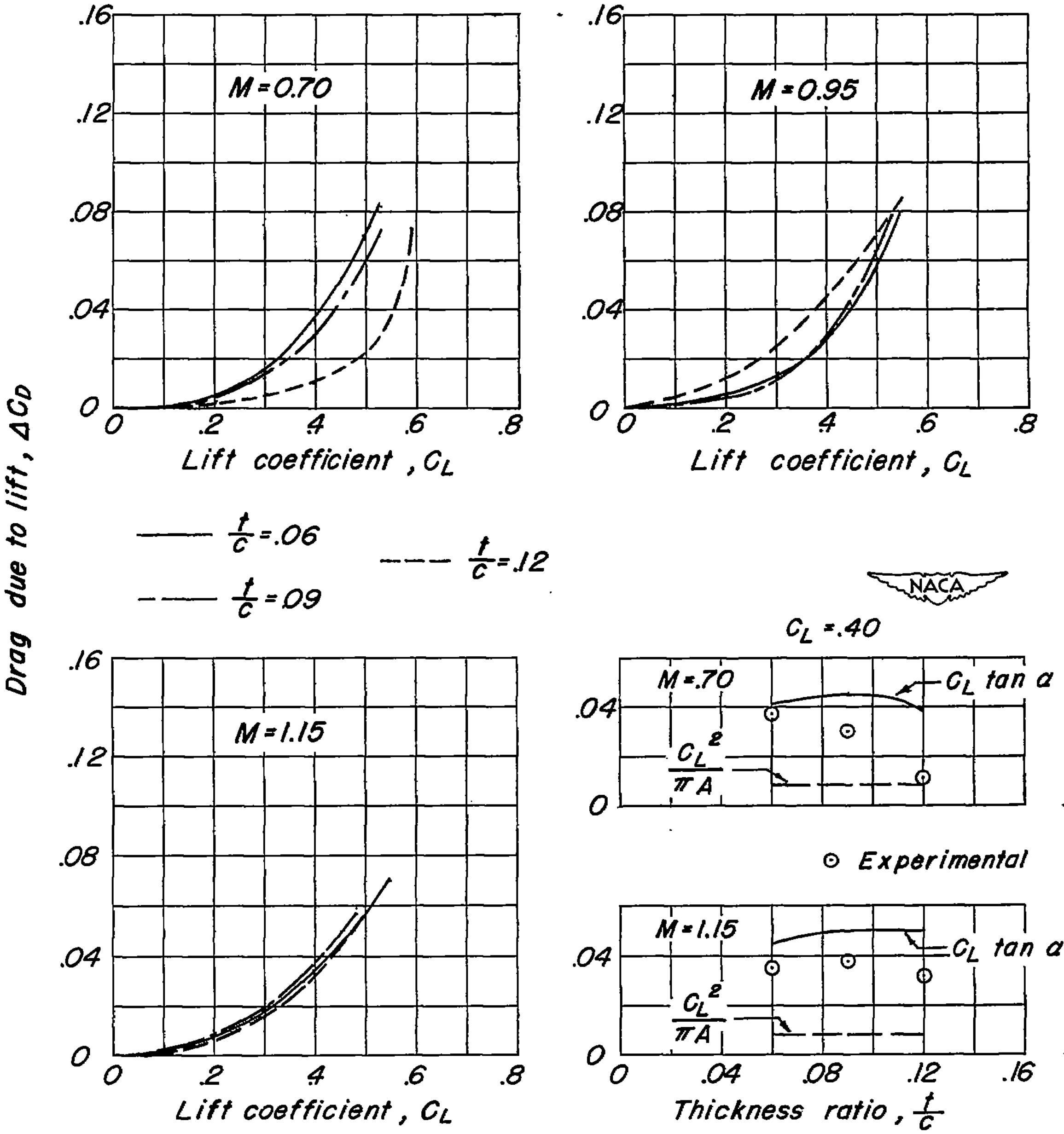


Figure 12.- Effect of thickness ratio on the drag due to lift. A = 6; $\Lambda = 45^\circ$.

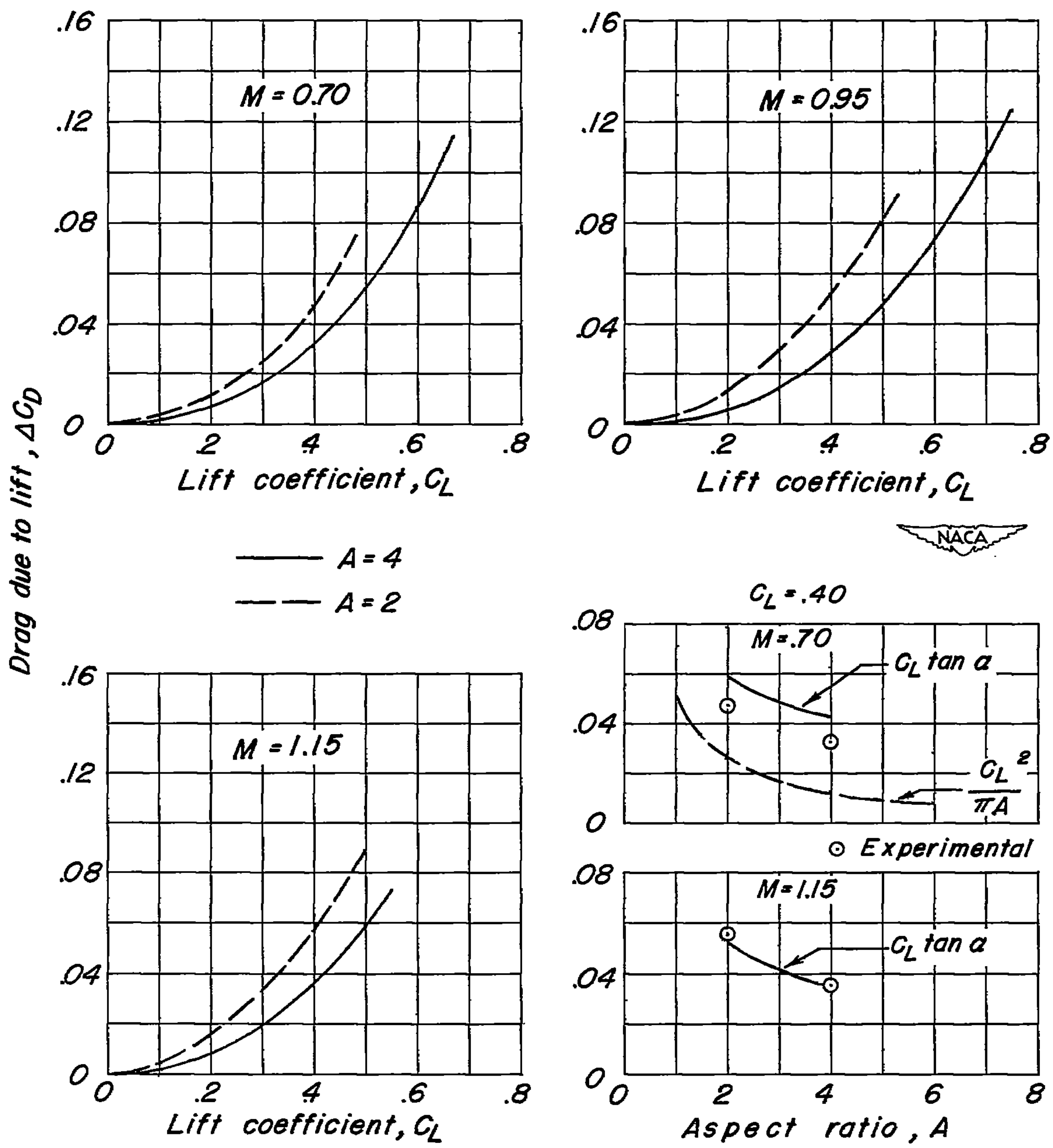


Figure 13.- Effect of aspect ratio on the drag due to lift. NACA 65A004 airfoil; $\Lambda = 0^\circ$.

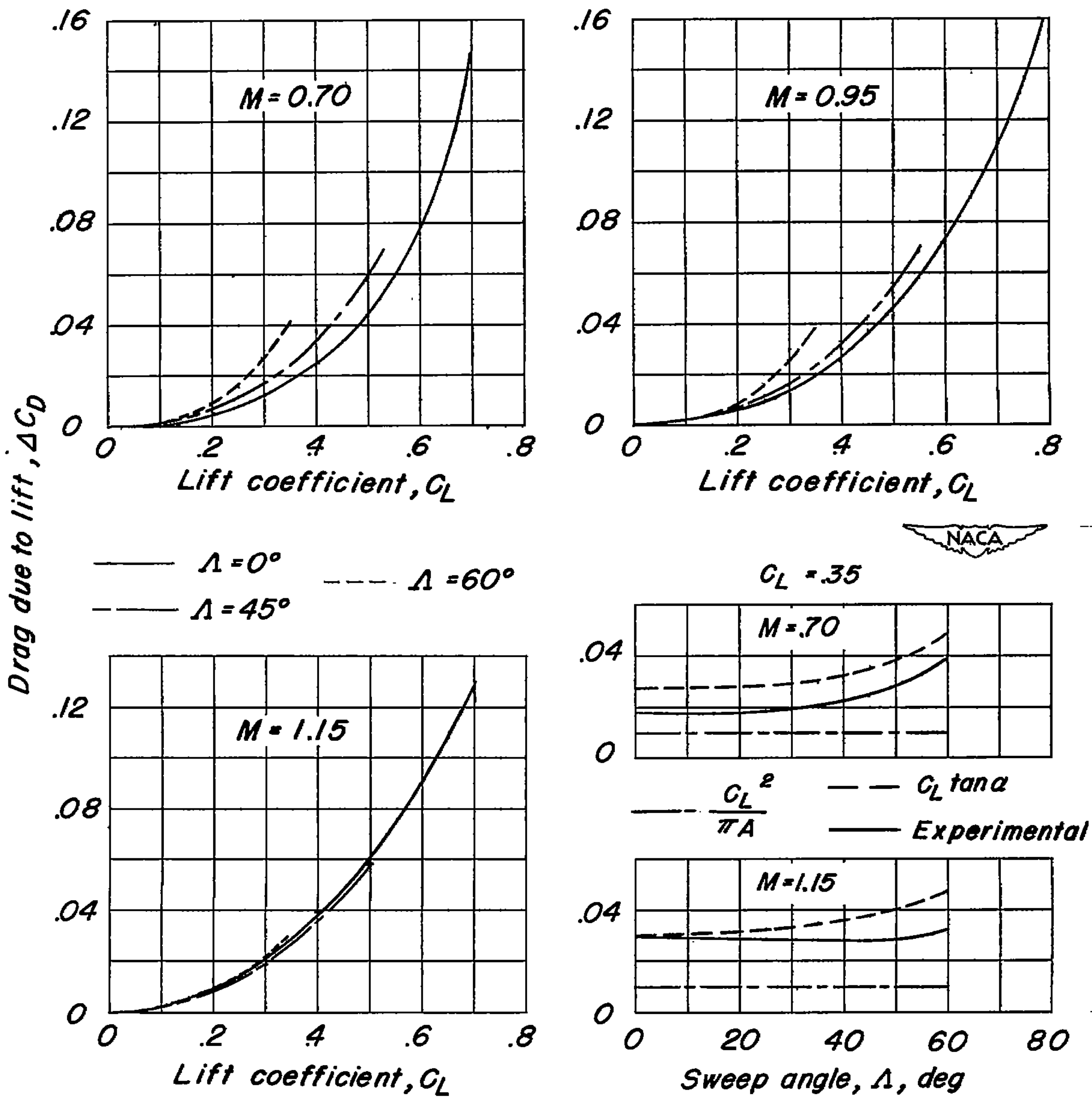


Figure 14.- Effect of sweep angle on the drag due to lift. A = 4; λ = 0.6; NACA 65A006 airfoil.

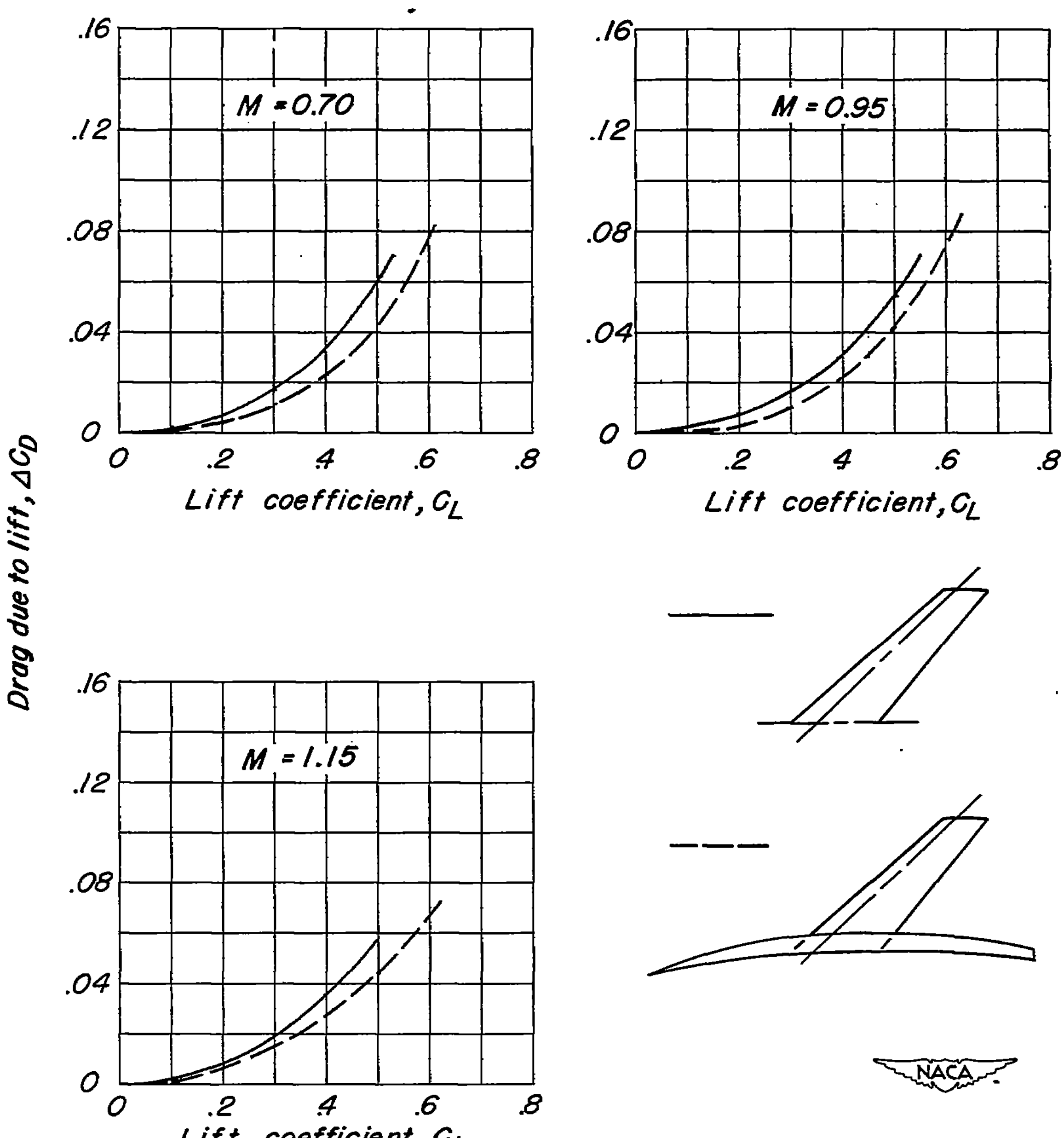


Figure 15.- Effect of fuselage on the drag due to lift. A = 4; $\Lambda = 45^{\circ}$; NACA 65A006 airfoil.